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(NASA-CR-110263) SPACE SHUTTLE PROPULSION ISSUE, STAGED COMBUSTION BELL VERSUS TAP-OFF OR GASGENERATOR AEROSPIKE /U/ (Bellcomm, Inc.) 65 p
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TECHNICAL MEMORANDUM

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(NASA-CR-110263) SPACE SHUTTLE PROPULSION ISSUE, STAGED COMBUSTION BELL VERSUS TAP-OFF OR GASGENERATOR AEROSPIKE /U/ (Bellcomm, Inc.) 65 p

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COVER SHEET FOR TECHNICAL MEMORANDUM

TITLE- Space Shuttle Propulsion Issue, Staged Combustion Bell Versus Tap-Off or Gas-Generator Aerospike (U)

TM- 70-1013-1

DATE- February 3, 1970

FILING CASE NO(S)- 103-8

AUTHOR(S)- C. Bendersky

FILING SUBJECT(S)- (ASSIGNED BY AUTHOR(S)-
Space Shuttle, Propulsion
ILRV, Propulsion
Space Transportation System, Propulsion

ABSTRACT

(U) This memorandum contains briefing charts and text assembled to compare a High Pressure Staged Combustion Bell Engine with a Tap-Off (or Gas-Generator) Aerospike Engine cycle for use on a LO_2/LH_2 recoverable Space Shuttle which would be operational in the late 1970's. It was concluded that the Hi-Pressure Bell Staged Combustion Cycle is better suited to the Space Shuttle Program.

(U) The material in an abbreviated form was presented at the Space Shuttle Design Criteria Review on October 18, 1969.

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FIGURE 15c CONTAINS SUBJECT MATTER COVERED BY A SECRECY ORDER WITH A MODIFYING "SECURITY REQUIREMENTS PERMIT" ISSUED BY U. S. COMMISSIONER OF PATENTS.

BA-145A (8-68)

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SUBJECT: Space Shuttle Propulsion Issue, Staged Combustion Bell Versus Tap-Off or Gas-Generator Aerospike (U) - Case 103-8

DATE: February 3, 1970

FROM: C. Bendersky

TM: 70-1013-1

TECHNICAL MEMORANDUM

(U) This memorandum contains the briefing charts and text assembled to address the issue of Space Shuttle main propulsion engine selection between two competing engine cycles, the Staged Combustion High Pressure Bell, and the Tap-Off (or Gas-Generator) Aerospike. The subject was addressed in terms of a LO_2/LH_2 recoverable vehicle operationally in use in the late 1970's.

(U) The rationale used for the paper was to assume that both concepts were technically feasible and then to develop arguments on relative costs, risks and payoffs in terms of vehicle configuration and engine development trades. It was concluded that the Staged Combustion High Pressure Bell engine cycle is more suitable than the Aerospike for the time frame of the late 1970's. The Bell engine technology is better understood, has apparently less development risk and the engine design is less closely coupled to the vehicle configuration.

(U) The memorandum contains 18 primary figures and 12 backup figures. These include data on the engine cycle characteristics and available supporting technology. The enclosed material in an abbreviated form was presented at the Space Shuttle Design Criteria Review (DCR) at NASA Headquarters, October 18, 1969.

(U) The interest and assistance of Dr. E. W. Hall/MTG in this study is greatly acknowledged. Appreciation is also tendered to A. O. Tischler/RP and W. W. Wilcox/RPX for their stimulating critiques. However the thoughts of this report are the considered opinion of this writer and do not necessarily represent those of the three mentioned.

C. Bendersky

C. Bendersky

1013-CB-baw

FIGURE 15c CONTAINS SUBJECT MATTER COVERED BY A SECRECY ORDER WITH A MODIFYING "SECURITY REQUIREMENTS PERMIT" ISSUED BY U. S. COMMISSIONER OF PATENTS.

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LIST OF FIGURES

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PROPULSION ISSUE

SPACE TRANSPORTATION SYSTEM
(SPACE SHUTTLE)

STAGED COMBUSTION BELL

VS.

TAP-OFF OR GAS-GENERATOR
AEROSPIKE

~~CONFIDENTIAL~~

[redacted]

ISSUES (U)

(U) This briefing is concerned with the issues in selection between a staged combustion bell and a tap-off (or gas-generator) aerospike propulsion system in terms of usage on a fully recoverable Space Shuttle (Space Transportation System) which is operational in the last half of the 1970 decade.

(U) The question of feasibility will not be addressed as there is no reason to believe that either concept could not work. The presentation will compare the concepts by development of arguments for relative costs, risks, and payoffs. The discussions will focus on vehicle configuration and engine development issues. Data are also presented on the concept backgrounds and technology differences.

(U) Detail backup charts are appended to amplify points. These are organized by the figure number; thus 12a, 12b, etc. are backup figures for figure 12 etc.

ISSUES

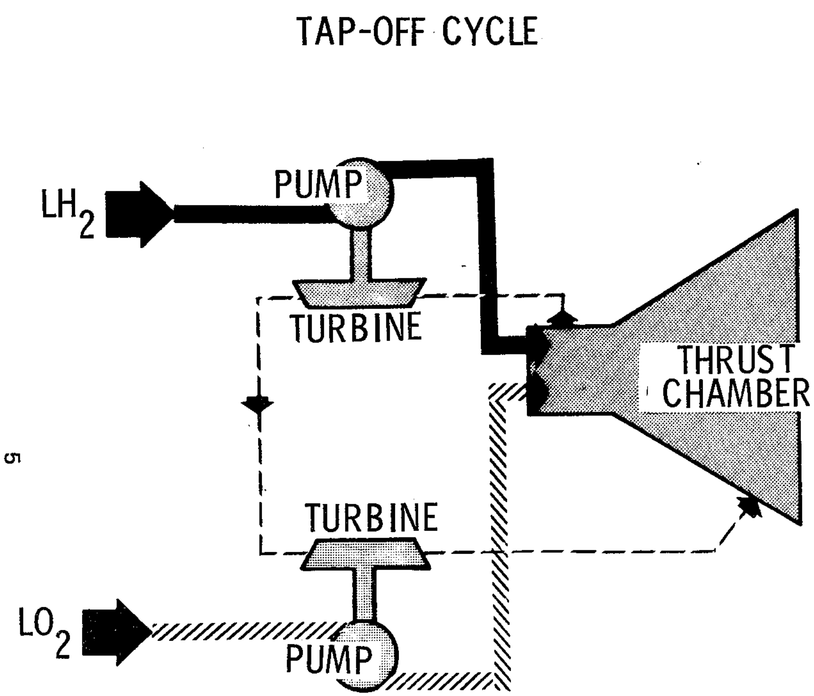
- SPACE SHUTTLE ONLY..
  - 1970'S OPERATION
- ASSUME BOTH CONCEPTS FEASIBLE
- DEVELOP ARGUMENTS ON...
  - COST • RISKS • PAYOFFS
- FOCUS DISCUSSION ON...
  - VEHICLE CONFIGURATION &
  - ENGINE DEVELOPMENT ISSUES
- INCLUDE
  - CONCEPT BACKGROUND & TECHNOLOGY DIFFERENCES
- DETAIL BACKUP CHARTS AVAILABLE

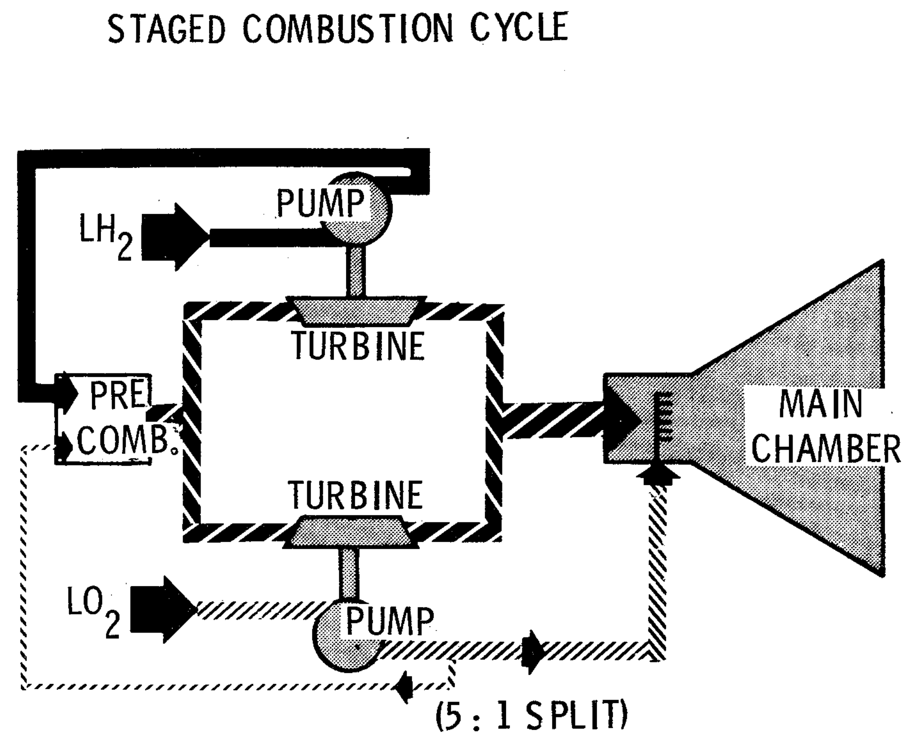
CYCLE SCHEMATICS (U)

(U) In the tap-off cycle, the Isp is a weighted average of the high energy combustion gases and the lower energy turbine exhaust gases. Turbine power to drive the high pressure ratio turbines is obtained from a small quantity of tapped-off, low temperature, main combustion gases (<3%). The low energy turbine exhaust gases are ducted overboard in a convenient manner. The technique shown is that of the J-2 and J-2S, in which turbine exhaust gases are injected into the divergent (supersonic) portion of the nozzle to minimize Isp losses. The turbomachinery and thrust chamber performance are coupled because the tap-off gas properties are necessary for turbine design. Therefore the development of these components must also be coupled. By reverting to the J-2 type gas-generator cycle in which turbine power gases are supplied from a separate gas-generator combustion device, the component dependence is somewhat eliminated. However the penalties are the requirement for an additional high performance component and a substantial increase of engine complexity.

(U) In the staged combustion cycle, all the Isp is obtained from the high energy main combustion gases. The pump discharge high pressure LO_2 is split into two flows. Approximately 20% is ducted to the preburner where it is burned with all the pump discharge LH_2 and, thus energized, serves as the drive fluid for turbine power. The large volume of gases are used in low pressure ratio turbines and ducted into the main combustor where the $\approx$80% remaining LO_2 is added and combusted. Thus all the propellants are used at the maximum energy level. The cycle is such that major component performance is decoupled thus allowing for independent component design and development. Clearly the main chamber is completely decoupled and it can be shown that precombustor and turbomachinery efficiency affect chamber pressure (Pc) only and have but a secondary effect upon Isp.

CYCLE SCHEMATICS





KNOWLEDGE OF TAP-OFF GAS PROPERTIES NECESSARY FOR TURBOMACHINERY DESIGN

- MAJOR COMPONENTS' FLOW DECOUPLED
- COMPONENT DESIGN INDEPENDENT
- PRE-COMBUSTOR & TPA EFFIC. AFFECT CHAMBER PRESSURE ONLY: NOT I_{SP}

ROCKETDYNE AEROSPIKE AND PRATT & WHITNEY STAGED COMBUSTION SCHEMATICS (U)

(U) The Aerospike proposed by Rocketdyne uses the particular type of tap-off or gas-generator cycle in which high performance combustion takes place in an annular main chamber consisting of several (or many) independent segments. The combustion gases flow through an annular throat onto an inner expansion surface whose length is typically 15-25% of a 15° cone having the same expansion ratio (ε). Shown is the dual chamber concept in which there are two rings of annular combustion chambers. Each ring is operated as an independent thrust chamber supplied either by a single or dual set of pumps. The ε is defined by the circular diameter of the annulus (De) and the throat flow area (A_t) ($\varepsilon = \Pi De^2/4A_t$).

The nozzle is altitude compensating as the combustion gases are free to expand and assume the profile consistent with the ambient pressure. Turbine exhaust gases are ducted into the base and under proper conditions of flow and altitude provide a small but important thrust increase. These turbine exhaust gases are termed "base bleed."

(U) The staged combustion engine system proposed by P&W is basically as shown. The engine is provided with a two-position nozzle, shown here in the retracted sea-level position. At altitude the nozzle is extended to provide high vacuum ε. Altitude compensation is achieved by the two positions in combination with the high Pc.

(U) Aerojet has proposed a variation of the staged combustion cycle having two major differences:

(1) The LO_2 and LH_2 turbopumps each have a separate precombustor and

(2) Two LO_2 pumps are used; the larger pump delivers all the LO_2 to the pressure required for the main chamber; the smaller pump provides the extra pressure for the LO_2 precombustor flow.

ROCKETDYNE AEROSPIKE & STAGED COMBUSTION SCHMATICS

AEROSPIKE

SOURCE:
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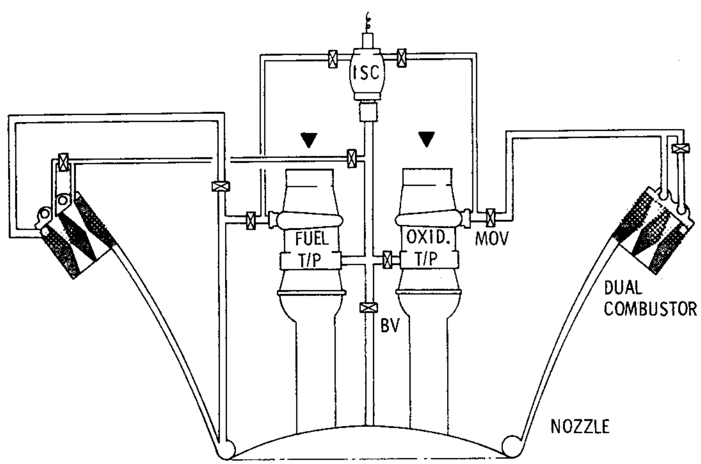
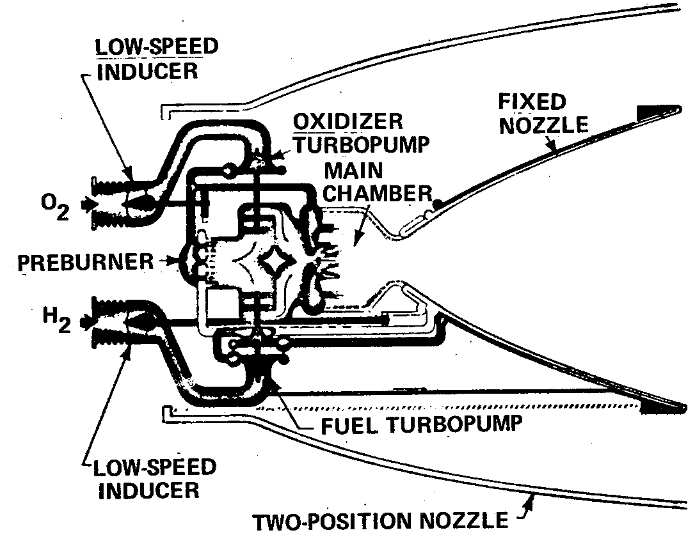


ALTITUDE COMPENSATING

P&W STAGED COMBUSTION

SOURCE:
P & W AIRCRAFT

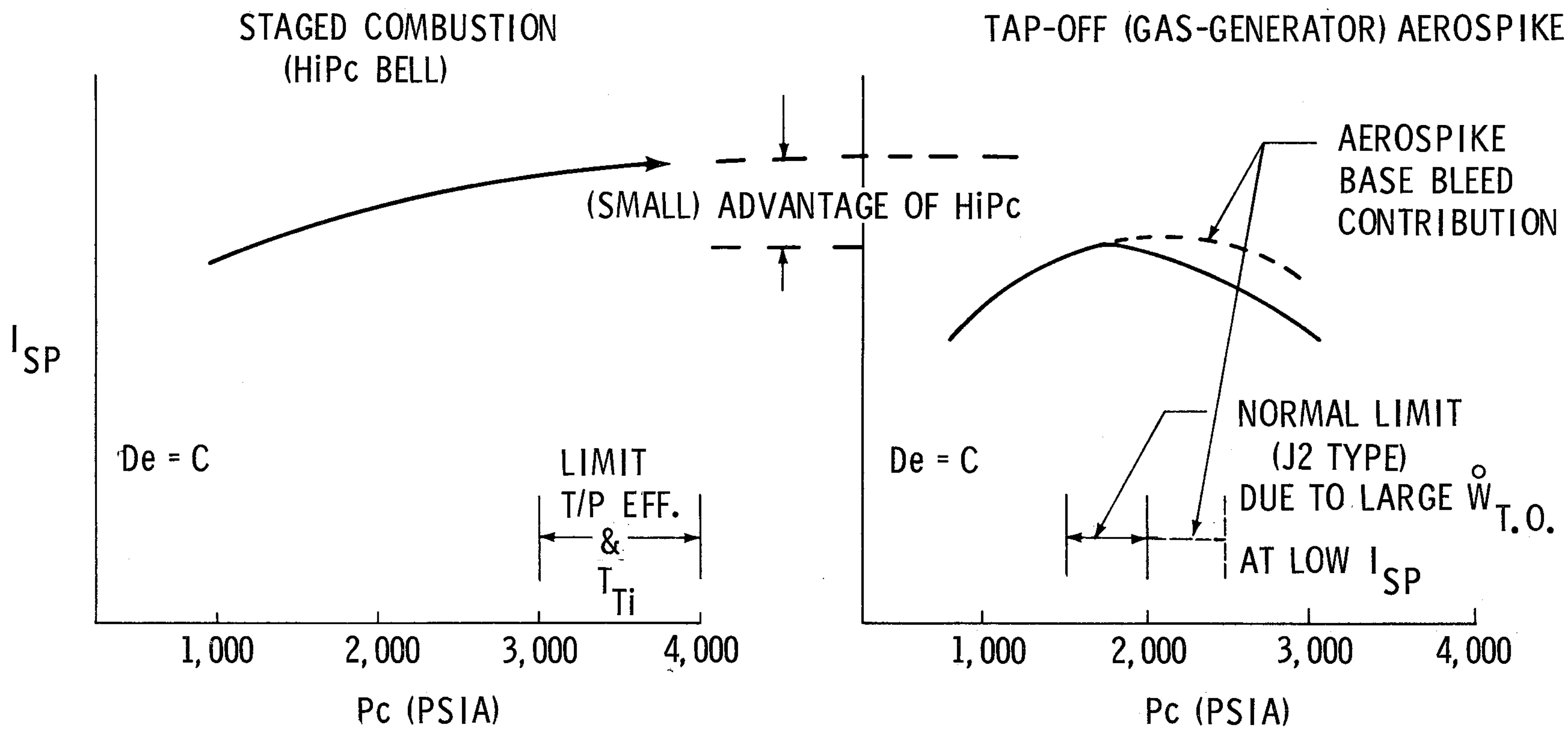


CYCLE PERFORMANCE (U)

(U) The staged combustion cycle "Hi Pc" bell engine exhausts all its combustion products at main chamber enthalphy, and for fixed exit dimension (De) higher chamber pressure (Pc) results in an Isp increase. The turbomachinery efficiency and allowable turbine inlet temperature proscribe a practical maximum Pc limitation which lies in the 3000-4000 psia range.

(U) The tap-off (or gas-generator) cycle requires low energy turbine gases in a quantity roughly proportional to the Pc. The system Isp is a weighted average of the main combustor and turbine gas flows. Thus an Isp maximum exists as a function of Pc and turbine flow. For a J2 type engine the cycle Isp maximizes in the 1500-2000 psia Pc range. The aerospike base bleed can significantly increase the Isp contribution of the turbine gases and thereby shift the Pc maximum point in excess of 2500 psia. Nevertheless, for a given De, the staged combustion cycle always has a small but significant 5-to-15 seconds Isp advantage. However, since an aerospike nozzle is shorter than a bell nozzle, the importance of this performance for length limited applications difference is reduced or may disappear. In a dual combustor configuration the aerospike may out perform a bell at altitude (at a lower thrust) with one combustor ring shut-off. This mode effectively increases the expansion ratio by reducing the throat area. Both previous statements imply that the ε increase exceeds the effect of the higher Pc of the staged combustion cycle.

CYCLE PERFORMANCE



PURE CYCLE BALANCE

- HiPc ALWAYS HIGHER THEO. I_{SP} THAN A/S FOR DIAMETER LIMITED APPLICATIONS
- HOWEVER LENGTH LIMITED CONFIGURATION COULD HAVE A/S I_{SP} > HiPc
- A/S DUAL COMBUSTORS CAN HAVE I_{SP} > HiPc AT ALTITUDE

PROGRAM HISTORY (U)

(U) Both NASA and the Air Force (A/F) have funded technology programs on both cycles with both cryogenic and earth storable propellants, starting in the early 1960's. For LO_2/LH_2, Rocketdyne has worked with the aerospike and Pratt & Whitney the staged combustion cycle. Through 1969, approximately $50 million has been spent at each company on these programs. NASA concentrated its support toward technology in the 350K lb thrust size in the advanced engines aerospike/bell programs (AEA/AEB) while the A/F has chosen a 250K lb thrust size advanced development program (ADP). In the mid 1960's the A/F sponsored an ADP hardware competition at the 250K thrust level. Pratt and Whitney was selected the winner based on applications studies, hardware tests and technology costs. The Advanced Cryogenic Rocket Engine (ACRE) subpanel of the AACB/SSRT had agreed that the A/F should take their preference and that NASA should fund the other cycle. Due to NASA (OART) funding limitations this intention was not implemented.

(U) Thus the only major high performance LH_2/LO_2 engine work underway is that at Pratt & Whitney and is called the "Reusable Rocket Engine Program (RREP)." The RREP objective is a feasibility demonstration of the 250K thrust engine*, with two-position nozzle in a flight type (not flight weight) configuration.

---

*This engine is identified by the Air Force model number XLR129P-1.

PROGRAM HISTORY

- AIR FORCE ADP & NASA AEA, AEB

  A/F ≈ 250K (F)   OART ≈ 350K (F)

- ADP 250K HARDWARE COMPETITION

  P & W SELECTED

  - APPLICATIONS STUDY
  - PERFORMANCE & COMBUSTION TESTS
  - TECHNOLOGY COST

  ACRE PANEL : NASA TO FUND A/S

- AF RREP PROGRAM TO P & W
  (XLR 129-P-1 DEMONSTRATOR ENGINE)

11

POTENTIAL ADVANTAGES OF THE AEROSPIKE (U)

(U) The potential advantages of the aerospike stem from features of shorter length, structural integration into the vehicle base, cheap combustor development and less turbomachinery risk.

(U) The smaller length allows a gimballed engine to make full use of the base area and satisfy the dynamic envelope established by gimbal angle requirements. The effect is to increase De and hence ε and Isp. In addition the smaller length may allow a superior base configuration and engine protection system during reentry (main engines off).

(U) In an integrated configuration, the aerospike combustor is shaped to conform to that of an optimum vehicle base (i.e., oval, 2-dimensional strip) best utilizing the base area. The integration can result in higher ε and Isp, reduced thrust structure weight, and body base drag, the latter of interest during reentry flight.

(U) The cheap development accrues through the "segment test concept". A segment is a complete independent portion of a thrust chamber (or button). By extensive pressure-fed segment testing a large reduction of full scale testing of the multi-segment module and the full scale pump-fed engine is contemplated. This would result in a shorter overall development time and the attendant savings in hardware, facilities, and propellant support.

(U) The validity of these potential advantages, in context of an early development program is addressed in the following discussions.

POTENTIAL ADVANTAGES OF AEROSPIKE (U)

SHORTER LENGTH

- SMALLER DYNAMIC ENVELOPE (HIGHER ϵ)
- LESS ENGINE &/OR BASE PROTECTION DURING REENTRY

STRUCTURAL INTEGRATION INTO VEHICLE

- BETTER BASE AERO CHARACTERISTICS
- HIGHER I_{SP} THROUGH MAXIMUM USE OF EXIT AREA
- LIGHTER VEHICLE WEIGHT

QUICK CHEAP DEVELOPMENT

- CHAMBER SEGMENT "QUAL"
- MULTI-SEGMENT MODULE
- BUTTON ENGINE OR PUMP-FED MODULE

AEROSPIKE CHRONOLOGY (U)

(C) Since 1960, Rocketdyne has conducted a large, extensive group of research programs on aerospikes which included cold flow, hot firings, heat transfer and configuration design phases. These programs have proven the predictability of theoretical nozzle design parameters. Combustion tests have been made with LO_2/LH_2, LF_2/LH_2, and earth storable propellants. Injector performance and limits of heat transfer (on a segment scale) are now known fairly well. Tests have been performed in tubular construction thrust chambers of 100-in. diameter but at less than design conditions. The cost of this type of chamber hardware was excessive. In 1968 Rocketdyne initiated testing of milled channel and cast segments and now is funded by OART in a cast channel chamber segment feasibility program. These concepts are designed to significantly reduce R & D and production engine costs and are considered important with regard to continued interest in the cycle.

(U) There are presently no funded LO_2/LH_2 engine type aerospike experimental programs.

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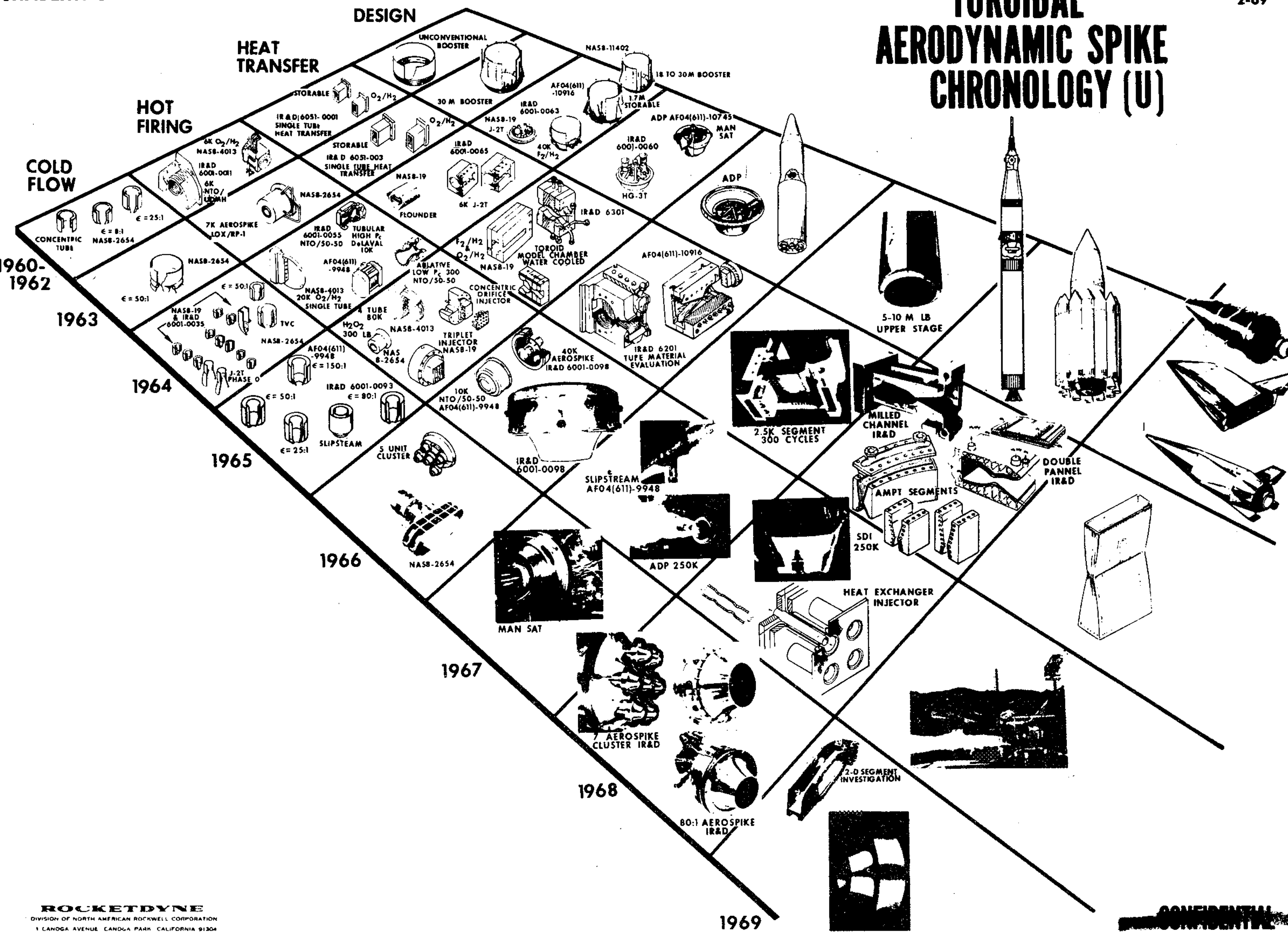
TOROIDAL AERODYNAMIC SPIKE CHRONOLOGY (U)



15

ROCKETDYNE
DIVISION OF NORTH AMERICAN ROCKWELL CORPORATION
CANOGA AVENUE CANOGA PARK CALIFORNIA 91304

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250 K THRUST AEROSPIKE CHAMBER FIRINGS (U)

(C) NASA and the Air Force funded construction and testing of pressure-fed LH_2/LO_2 circular (button) aerospike thrust chamber hardware. Shown is the sea level firing of a 100-in. dia. regeneratively cooled tubular thrust chamber. Base bleed is provided by a separate gas-generator. The chambers were designed for a Pc of 1500 psia to provide 250,000 lbs thrust at mixture ratios up to 7. In the programs, chamber pressure testing was limited to 1100 psia at mixture ratios from 3.5-to-5.0. A non-flight type (hypergolic CTF slug) ignition system was used. The chambers were over-cooled to conserve hardware and Isp performance was about 10 seconds (2 1/2%) lower than those now predicted for an optimized system. Pressure-fed segment testing has been successfully accomplished at up to 2700 psia Pc levels.

SOURCE:
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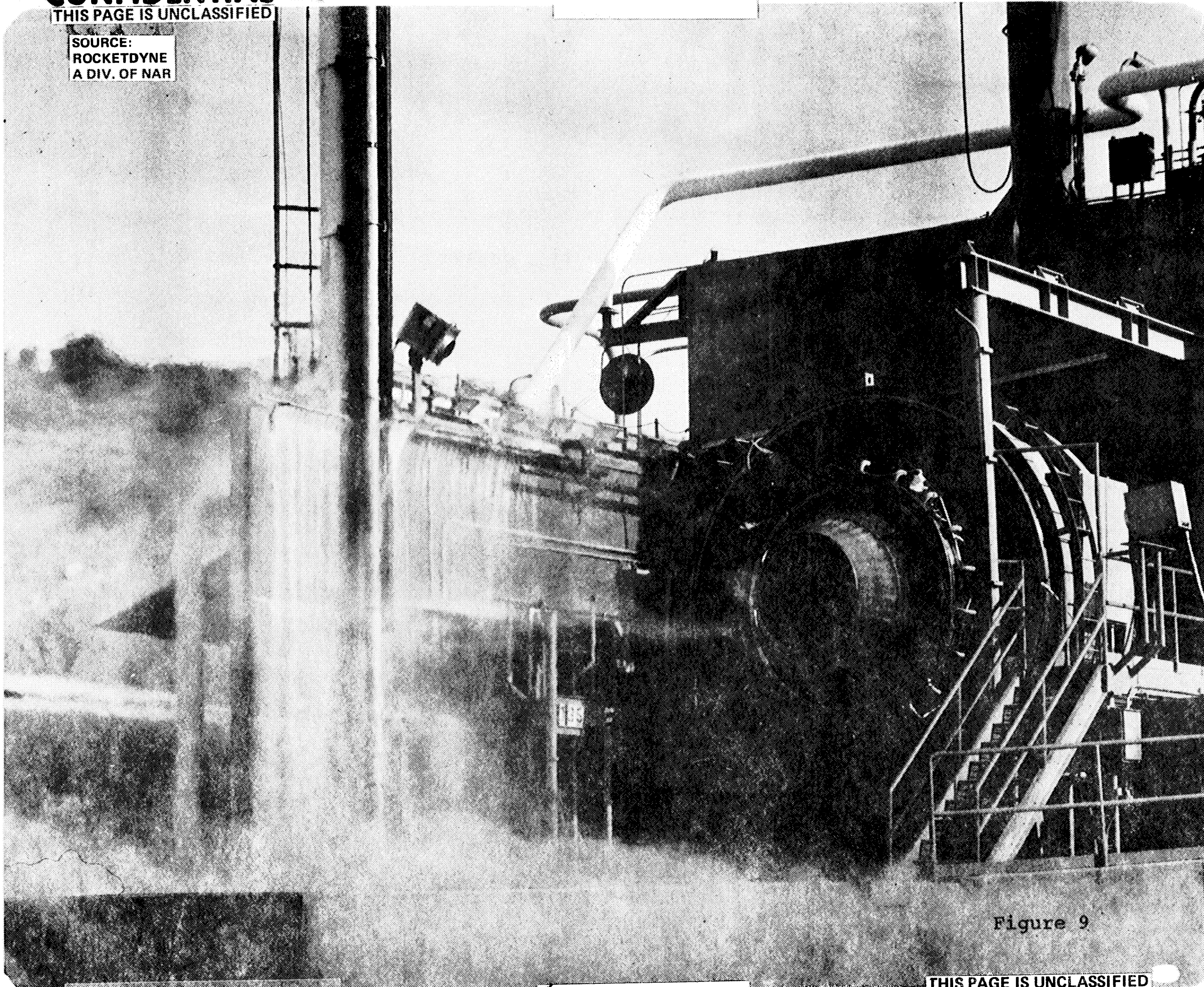

Figure 9

XLR129P-1 ENGINE CUTAWAY (U)

(C) The figure shows a cutaway of the RREP XLR129P-1 "Demonstrator" Engine. The engine assembly has not been fired. The callouts are those of the main hardware components. The turbopumps and combustors will be connected by a manifold called a transition case (not called out) which is a highly stressed structure designed to duct hot precombustor gases as well as accept the thrust and structural loads. This engine is designed with a gimbal mount to permit use of the main thrust vector for vehicle attitude control. The precombustor, primary nozzle, and transition case will be regeneratively cooled with LH_2. The high heat flux main chamber will be transpiration cooled, that is the chamber walls will be separated from the combustion zone by GH_2 which is injected (transpires) through grooved copper wafers. The two position nozzle will be LH_2 "dump" cooled, that is a small amount of required LH_2 coolant will be exhausted overboard after a single pass.

(C) The XLR129P-1 "Demonstrator" program is scheduled for completion in 1972. It will include a series of 30 engine firings to demonstrate 95% Isp efficiency, 5:1 throttling, 5-7 mixture ratio operation, and a potential TBO (time between overhaul) of 10 hours, and 300 restarts.

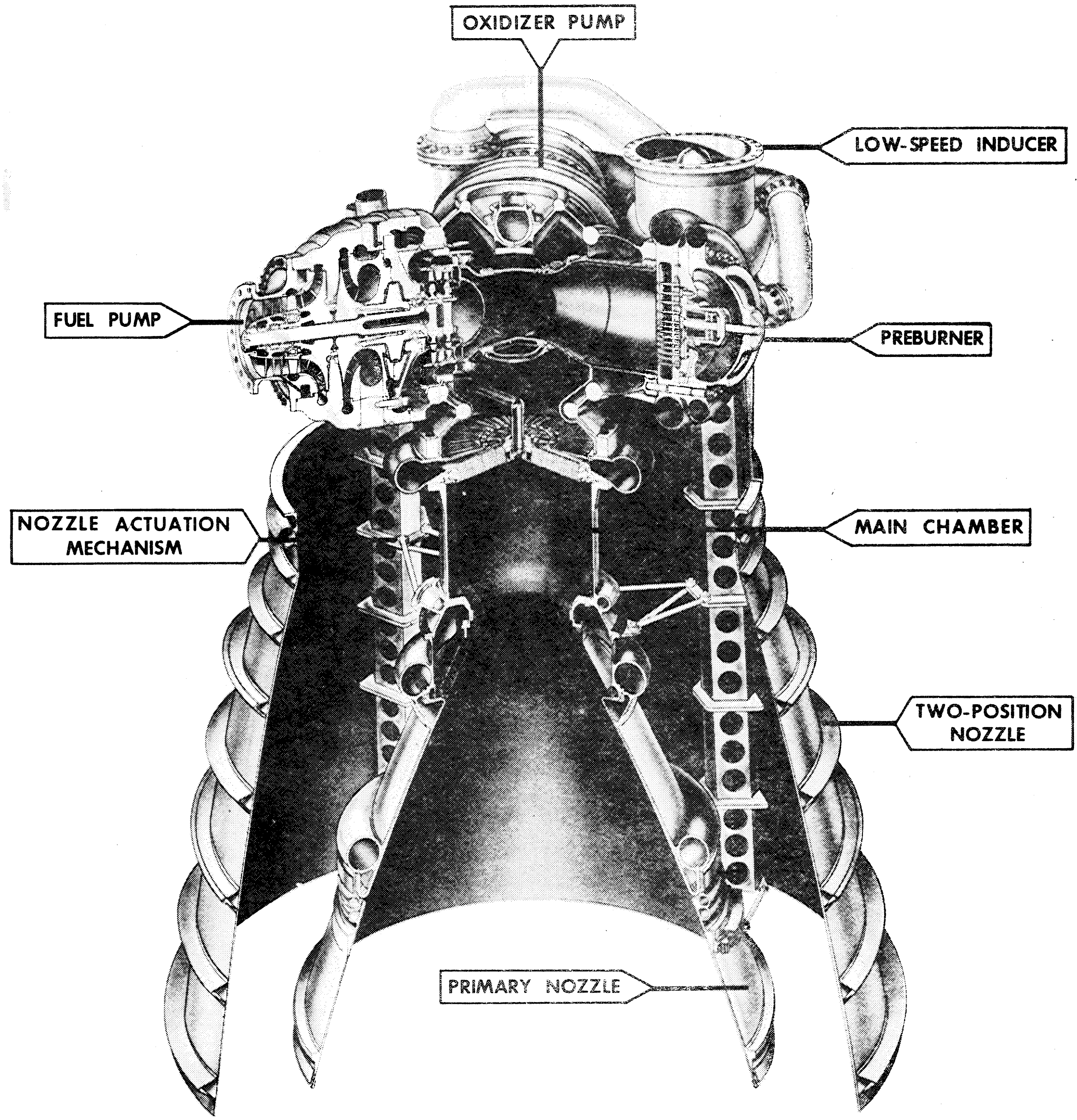
OXIDIZER PUMP
LOW-SPEED INDUCER
FUEL PUMP
PREBURNER
NOZZLE ACTUATION MECHANISM
MAIN CHAMBER
TWO-POSITION NOZZLE
PRIMARY NOZZLE

Figure 10

STAGED COMBUSTION LO_2/LH_2 Hi Pc HISTORY (U)

(U) Shown is the high pressure LO_2/LH_2 hardware used to support the RREP and its schematic relation to the XLR129P-1 "Demonstrator" Engine. (The engine assembly has not as yet been fired). All components except the transition case are working hardware, and are available for RREP support. The firing shown is that of a full scale 250 K lb thrust chamber. Both precombustor and main burners were tested at full scale and were ignited with flight type ignition systems similar to that of the J-2 & RL-10 Engines.

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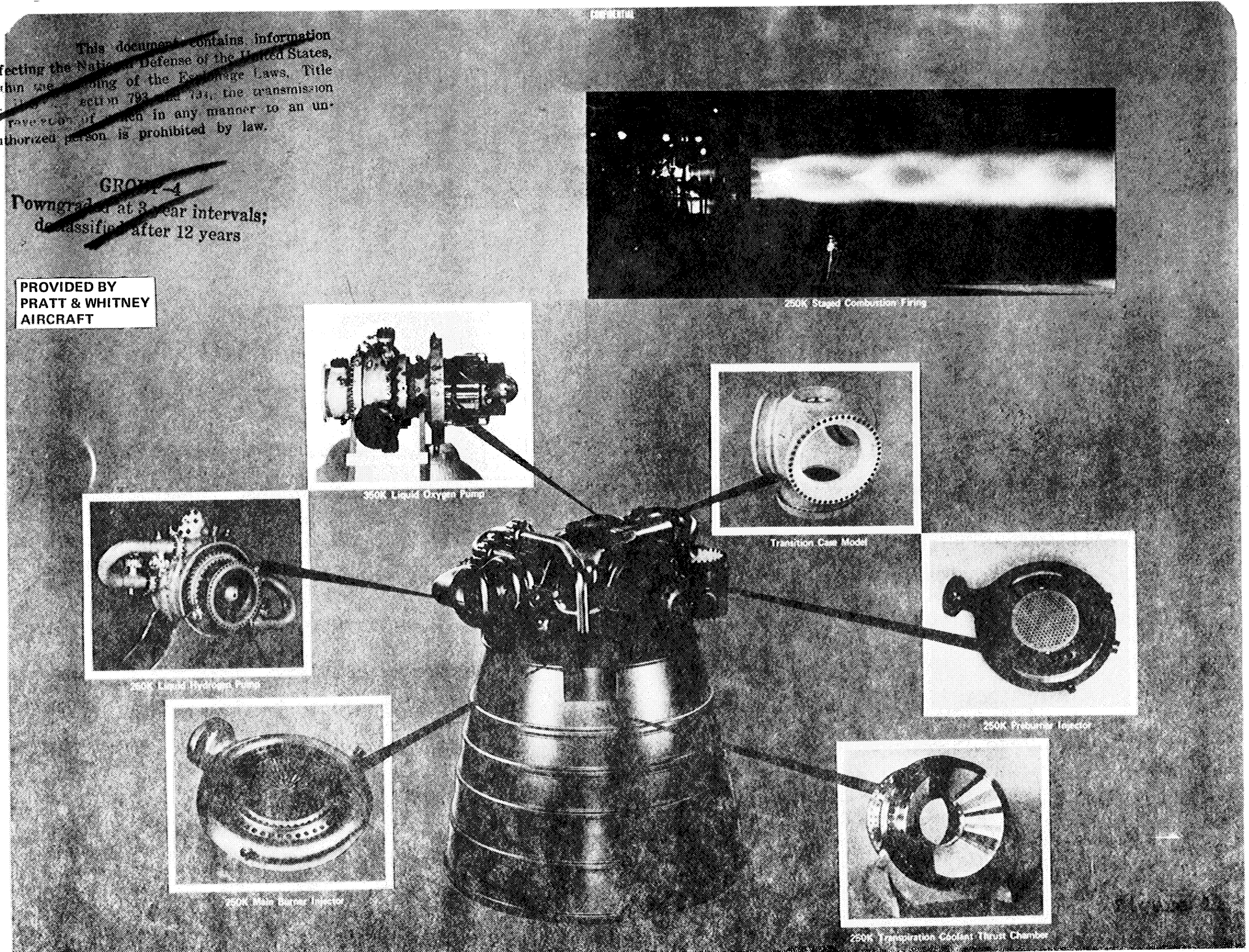

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ARES HISTORY (U)

(C) This chart shows the results of the Air Force funded ARES (advanced rocket engine, storable) program at the Aerojet General Corporation prior to redirection and deemphasis. This earth storable, 100 K lb. thrust, staged combustion engine concept was breadboarded and successfully fired as shown in the figure. A Pc of 2800 psia was obtained with flight type precombustor and main combustor.

(U) The program results are a verification of the staged combustion cycle performance potential.

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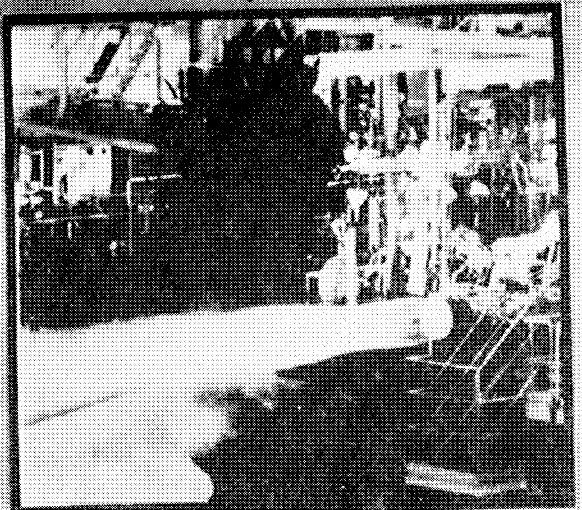
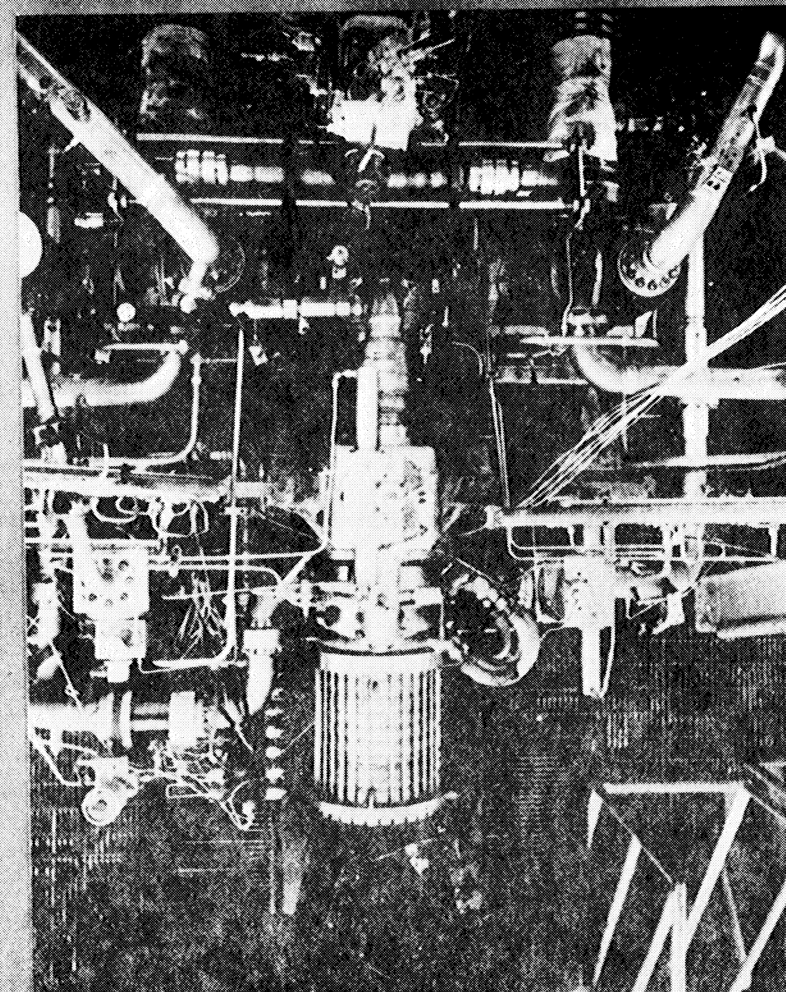
HIGH PRESSURE ENGINE TESTING (U)



TEST RESULTS

- 46 TESTS
- $I_s > 90\%$ (4 TESTS)
  (1 - 20 SEC)
- $I_s > 89\%$ (13 TESTS)
  (4 - 20 SEC)
- POTENTIAL 3% I_s IMPROVEMENT
- P_C=2800 PSIA



AEROJET GENERAL TIRE GENERAL ®

LIQUID ROCKET DIVISION

Figure 12

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SUMMARY: BASIC TECHNOLOGY DIFFERENCES (U)

(U) A list of key technology differences and critical issues are presented. Backup figures are appended on the items of Pc limit, ignition, turbine drive (tap-off) and one chart for structural, throttling and idle-mode issues. The above items are summarized as follows. Outside of the risks of the non-demonstrated staged combustion Hi Pc transition case, the technology risks for an aerospike compare unfavorably. The staged combustion problems are those historic to a development plan. Those of the aerospike require high risk early decisions. Errors in these decisions which will require costly (money & time) remedies.

(U) Many of the aerospike risks are concomitant with its cycle features and previous lack of exploratory funding. In addition, severe basic hardware drawbacks do exist in comparison to Hi Pc staged combustion engines. These are associated with the annular combustor structure, the large amount of segment/module/engine manifolds and propellant feed plumbing and generally less understood system aspects.

SUMMARY : BASIC TECHNOLOGY DIFFERENCES

	AEROSPIKE	STAGED COMBUSTION
CH'BER PRES. LIMIT (Pc)	HEAT TRANSFER	TURBOMACHINERY
IGNITION (SPARK PLUGS)	SEGMENTS	DUAL OR SINGLE TORCH
TURBINE DRIVE	SEGMENT TAP-OFF	PRIM. COMBUSTOR
STRUCTURAL	SEGMENTS/THROAT	TRANSITION CASE
THROTTLING	??	FIXED GEOMETRY
IDLE MODE	COMPLEX IGNITION	MAIN COMBUSTION ONLY
STABILITY	BAFFLES/SLOTS	INHERENTLY STABLE
THRUST VECTOR CONT.	LITVC OR HEAVY GIMBAL	CONVENTIONAL GIMBAL
TANK PRESS. H_2 GAS	COMPLEX	CONVENTIONAL
MANIFOLDS, PLUMBING, ECT.	COMPLEX	CONVENTIONAL
SAFETY, ENGINE OUT	???	S-II, S/C TYPE

SUMMARY: BASIC CONFIGURATION ISSUES (U)

(U) The impact of engine selection on the configuration issues, if any, is not apparent. It is not clear which concept will deliver higher Isp. However the RREP main combustor has demonstrated a satisfactory measured Isp performance; the aerospike has not as yet. Preliminary design weight comparisons do not reveal a firm superiority for either type of engine. Engine influenced vehicle base drag and c.g. shift data (figure 14A) similarly do not provide a basis for propulsion selection.

(U) Engine installation considerations for comparing clustered Hi Pc bell or clustered single combustor aerospikes modules have not resulted in a significant selection criterion. However, if a dual combustor or a tailored base aerospike (figures 14b, 14c, and 14d) is used, the combustor segment shape, size, and packaging will be dependent upon and different for each particular vehicle configuration, even if the total thrust per module is constant. Thus a more detailed aft-end vehicle predesign would be required prior to aerospike design while the Hi Pc bell engine only requires the thrust level selection before engine go-ahead. Early flight testing may indicate the need of vehicle base modifications. The two-position skirt provides some flexibility in configuration modifications; the aerospike does not.

SUMMARY: BASIC CONFIGURATION ISSUES

	AEROSPIKE	HiPc BELL
PERFORMANCE (I_{SP})	? ? ?	DEMONSTRATED
WEIGHT	? ? ?	? ? ?
BASE DRAG	NOT AN ISSUE	NOT AN ISSUE
CG TRIM	NOT AN ISSUE	NOT AN ISSUE
PROPULSION SYS. OPTIONS	TAILOR BASE (MODULES) DUAL COMBUSTORS BUTTON CLUSTERED	CLUSTERED
ENGINE GO-AHEAD	AFTER PREDESIGN OF SELECTED CONCEPT(S)	AT CONCEPT(S) SELECTION
AFT-END TAILORING	VEHICLE ONLY	BOTH VEHICLE & ENGINE
ENGINE-ON PLUME EFFECTS	CONFIG. DEPENDENT	BETTER UNDERSTOOD
DYNAMIC TEST & POGO	? ? ?	BETTER UNDERSTOOD

SUMMARY: DEVELOPMENT ISSUES (U)

(U) Costs estimates for an engine development, historically, have been synthesized by specifying the type and number of full scale tests through PFRT and QUAL and then pricing out the engineering, hardware, facilities and propellants necessary for test plan execution. NASA will establish this test number. However historically PFRT require >500 tests and as many as 2000 are proposed for full shuttle engine qualification.

(U) The key to a cheap development is early full scale testing. Thus confidence that component performance can be maintained during full scale testing is essential. The XLR129P-1 program will, if accelerated, provide early verification of the engine system concept at a thrust level which is surely scaleable to full scale shuttle size (250 K to 1000 K thrust). The value and status of the program are shown in figures 15a, and 15b. Figure 15c shows the concept of assembly which amplifies the "independent" component aspect. It is judged that the aerospike experimental risks will probably necessitate a provision for testing recycle to the module or segment test level to verify component performance, during which full scale testing might have to be interrupted.

(U) Early availability infers rapid engine system reliability growth. The RREP critical items are presently under experimental evaluation to a greater degree than the aerospike.

SUMMARY : DEVELOPMENT ISSUES

- DEVELOPMENT COSTS α NO. TESTS AT ENGINE LEVEL
- XLR 129P-1 PROVIDES DIRECT BACKGROUND
  - HARDWARE ON WAY
- HiPc BELL HAS INDEPENDENT COMPONENTS
- MODULE HAS DEPENDENT COMPONENTS
  - A/S RISK HIGH DUE TO "RECYCLE TESTING"
- EARLY RELIABILITY GROWTH
  - HiPc BELL CRITICAL ITEMS UNDER STUDY
  - XLR 129P-1 COMPONENTS ASSIST EARLY EVAL.
  - J2-S PUMPS
- <u>CAN A/S BE CHEAPER THAN HiPc BELL ??</u>

STAGED COMBUSTION AND AEROSPIKE ENGINE DEVELOPMENT FLOW CHARTS (U)

(U) A typical development flow chart for a staged combustion engine is shown opposite, while that for an aeropsike engine (A/S) is on the next figure. Both refer to the same thrust level. The dashed box areas represent major component test phases. The circles inside are typical key parametric features which if arranged in a spoke infer concurrent evaluation.

(U) A primary deduced impact is that both concepts require many and extensive full scale component and engine systems tests. As such for an "all success" program differences in the numbers of full scale tests are small. Therefore development cost differences cannot be identified.

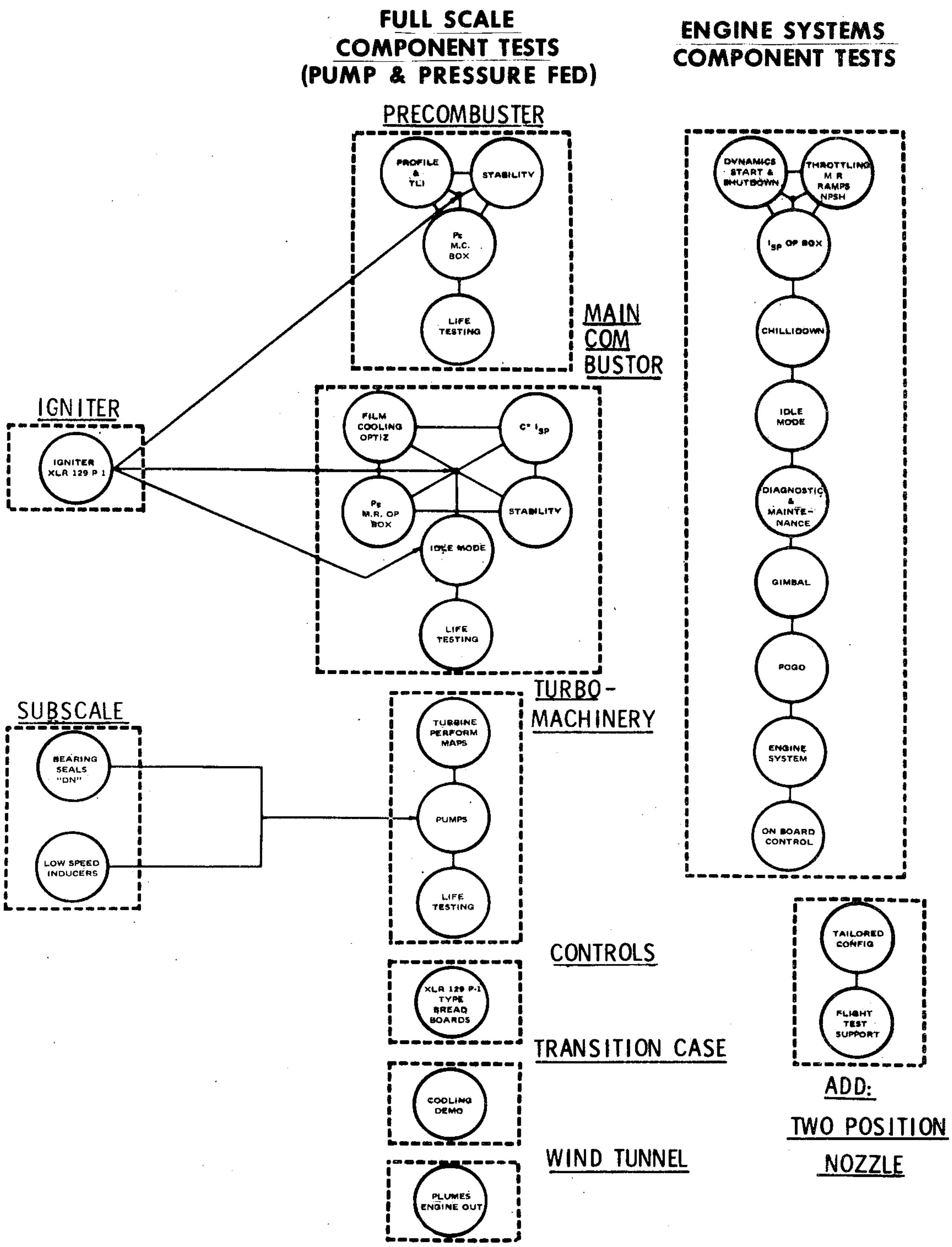
(U) The second impact is the effect of experimental feedback risks. The A/S segment screening allows a subscale evaluation of combustion performance. Two different segments are needed for the dual combustor. (The gas-generator A/S requires a separate high performance throttling combustion device.) The segments (say 100) must be evaluated as a button chamber assembly. Development problems will necessitate return to the segment test size, probably causing a delay in full scale button testing. After developing a segment hardware fix, 100 segments must be modified, assembled, and reevaluated. Similarly engine test problems may require recycling to the pressure-fed button or to the segment level. Engine system problems may therefore delay the necessary test time accumulation prior to vehicle support or flight test initiation. The dual chamber configuration multiplies the program risks significantly.

(U) On the other hand, staged combustion bell full-scale hardware testing can be scheduled concurrently with hardware and facility availability. Engine system testing can be initiated with derated combustion devices, probably as soon as turbomachinery becomes available. Thus key engine system problem areas can be evaluated in full scale tests concurrently with optimization of the combustion devices. Yet the engine cycle nature precludes engine systems-to-component development feedback. It is conceivable that derated hardware (say XLR129P-1 quality) will be available for vehicle development and flight operations very early in the program without danger that the qualified propulsion system will introduce different interface characteristics.

(U) Selection of an aerospike configuration tailored to the airframe base magnifies the development problems. For tailored base configurations such as figures 14b, 14c, or 14d, testing of the complete package (multi pump-fed modules) must be part of the propulsion development. Items such as priming, TVC, engine-out, and base bleed must be evaluated at vehicle full thrust, necessitating testing in a much larger facility. Development of tailored base A/S must be costlier than staged combustion engines or circular A/S modules.

(U) In all-success programs it is judged that both engine system costs are comparable. For early development, the higher A/S risk necessitates larger contingency allotments. Finally any development problems of the A/S may cause a large program slippage.

STAGED COMBUSTION FLOW CHART



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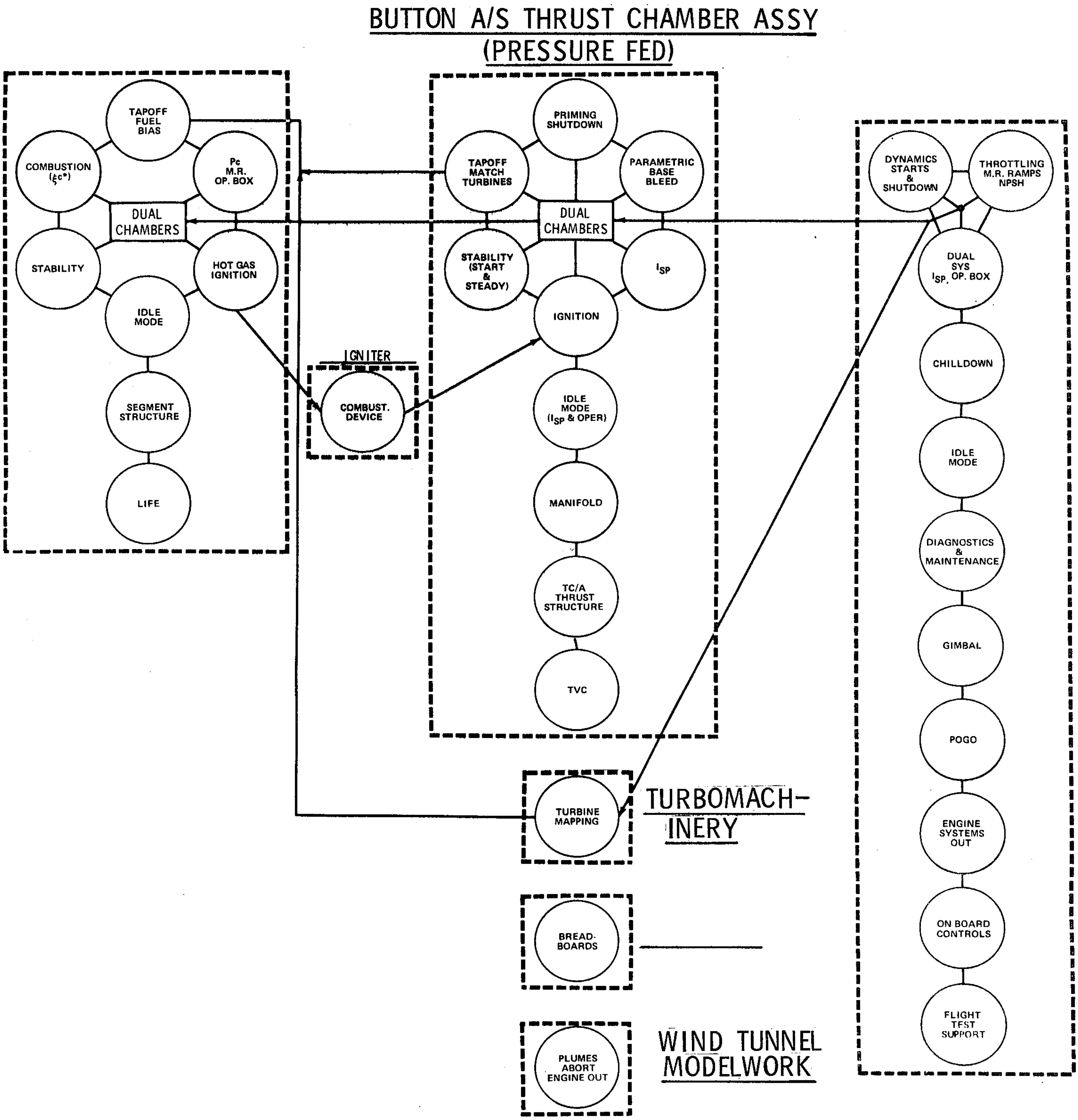
AEROSPIKE FLOW CHART

SEGMENT SCREENING | **FULL SCALE COMPONENTS** | **ENGINE SYSTEMS TESTS**



CONCLUSIONS (U)

(U) It is concluded that the staged combustion high pressure bell engine has a significant programmatic advantage over a tap-off (or gas-generator) aerospike engine in context of a main propulsion system for a 1970's space shuttle. These advantages stem from the inherent combustion and development features of the engine cycle, the maturity of the state-of-the-art in component development, and the continued existence of the Air Force funded Pratt & Whitney XLR129P-1 (Demonstrator) Engine Program. These facets provide a confidence that availability of the "critical path" main propulsion system will not unduly delay vehicle development and early flight operations.

(U) The value (if any) of potential aerospike advantages on system performance, configuration selection, or costs has not been established.

(U) The concluding statements are pertinent to a requirement for engines to support an early space shuttle only, and should not be interpreted to mean that aerospike technology is not a valuable and useful OART-type project.

CONCLUSIONS

- STAGED COMBUSTION HIGH PRESSURE BELL ENGINE OFFERS MORE.
    - EARLIER FINAL DESIGN
    - EARLIER AVAILABILITY
    - HIGHER PROBABILITY OF SUCCESS
    - NOT AS CONFIGURATION ORIENTED
    - NO DEFINED COST DIFFERENCE
    - NOT DEPENDENT ON NEW FABRICATION DEMONSTRATIONS
        - OART CAST A/S PROGRAM
        - DUAL COMBUSTOR

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BACK-UP CHARTS AND DISCUSSION (U)

(U) The following include supplementary information and substantiations for the information presented in the foregoing.

CHAMBER PRESSURE LIMITATIONS (U)
(Summary: Basic Technology Differences)

(U) The aerospike has an annular throat gap and large high heat flux surface areas. The gas side heat transfer coefficients (hc) are significantly higher than for a bell engine. For example at constant thrust, the same hc can be obtained at a Pc of 3000 psia with a bell engine as with an aerospike at 1750 psia. Since the aerospike is regeneratively cooled by H_2, the coolant pressure drop (ΔP) may become large which unfavorably shifts the aerospike power balance requiring operation at lower Pc and/or increasing the required turbine working gas flowrate. This penalty is increased with a requirement for emergency power capability.

(U) The staged combustion high heat flux main chamber is transpiration cooled for both Pratt & Whitney and Aerojet concepts. Transpiration cooling is decoupled from system pressure drops. The maximum chamber pressure is therefore limited by turbine life considerations and turbomachinery efficiency. Satisfactory (but low) pump performance for the shuttle engine has been demonstrated in NASA hardware contracts at the 350 K thrust level. OART (RP) states that present technology provides confidence that this performance can be improved from 6-9 percent.

CHAMBER PRESSURE LIMITATIONS (Pc)

AEROSPIKE

HEAT TRANSFER:

THROAT GAP

SURFACE AREA

RATES HIGHER (hc)

(3000 Pc BELL $\approx$ 1750 Pc A/S)

Δ P H_2 JACKET MAY CONTROL Pc MAX

OVERRIDE ON POWER BALANCE

ENGINE BALANCE MAY REQUIRE EXCESSIVE SECONDARY FLOW

125% MAX. POWER REQUIREMENT SEVERE

HiPc BELL

g/A_{MAX} LOW--FILM COOLED (< 1%)

TURBOMACHINERY LIMITED

350 K NASA PUMPS

	LOX	LH_2
Δ P (PSIA)	5670*	5682
EFFIC. (%)	68	66
DN (10^{-6})	1.36	2.28
THROTTLING		33.1

* BELOW TARGET M.R. TURBOMACHINERY CAN BE IMPROVED 6-9%

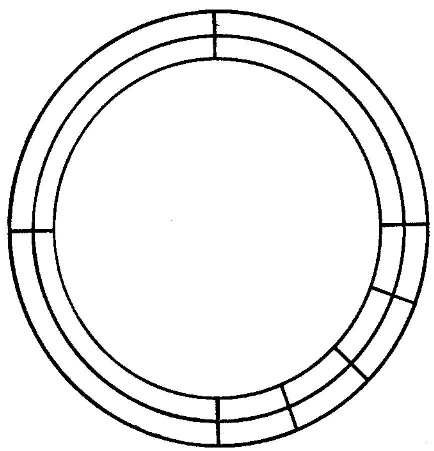
IGNITION (U)
(Summary: Basic Technology Differences)

(U) The engine ignition system must be capable of many hundreds of reliable restarts without major maintenance. Historically, for non-hypergolic propellants such as LH_2/LO_2, reliable ignition and reignition has been a major source of development and flight failures. The aerospike button or module is to be constructed of a large number of independently plumbed segments, each of which must have an independent ignition source. The presently preferred ignition concept uses hot gases (obtained from a separate combustor) ducted to each segment. Thus 50 to 100 small hot high pressure gas lines may be required. The staged combustion system at most requires 2 ignition systems, one each for the preburner and main burner. These hot gas "torches" are in use in the "RREP" and are similar to flight demonstrated units on the RL-10, J-2 and the SIV H_2/O_2 burner. Thus far no flight type ignition systems have been used in thrust chamber aerospike testing.

THIS CHART WAS TAKEN FROM A JUNE 1969 MSFC PRESENTATION TO DR. VON BRAUN

IGNITION

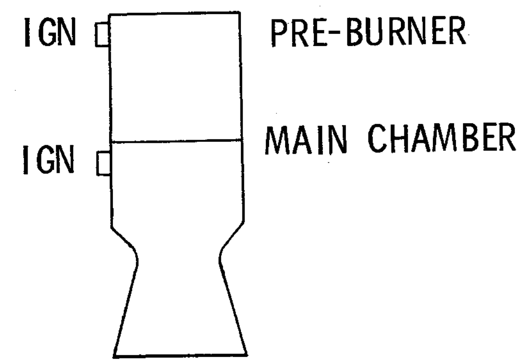
AEROSPIKE



IGN POINTS REQ'D 75
IGN METHOD - HOT GAS (SINGLE TORCH)
TECHNOLOGY STATUS:

- DEMONSTRATED IN SEGMENT TESTS
- 20 M. SEC. DELAY @ 1650^{0}F GAS

BELL



IGN POINTS REQ'D - 2
IGN METHOD - HOT GAS (2 TORCHES)
TECHNOLOGY STATUS:

- FLIGHT DEMONSTRATED
- RL-10
- J-2
- S-IV O_2/H_2 BURNER

SOURCE MSFC

TURBINE DRIVE GASES (U)
(Summary: Basic Technology Issues)

(U) The aerospike was originally proposed with a tap-off type turbine drive cycle, but a separate gas-generator turbine drive system is now favored. The cut and try aspects of tap-off, the necessity for multi-lines and manifolds to extract proper cool gases from each segment and duct them to the turbine, and the unknown tap-off effects on segment combustion and engine Isp have tended to disfavor the concept. The separate gas-generator eliminates the above problems but substitutes a component which, to maintain the proper amount of base bleed gases, must have a high combustion efficiency over the required wide range of engine operation.

(U) The nature of the staged combustion cycle does not require a high combustion performance in the precombustor. Rather the heated fuel rich turbine gases must provide combustion temperature profiles consistent with turbomachinery lifetime targets. Thus the development program will concentrate on the high thrust, high mixture ratio temperature profile. During throttling, a drop in combustion efficiency would be compensated by slightly increasing the precombustor LO_2 flow to reach the power balance temperature. The engine mixture ratio is maintained by decreasing the LO_2 flow to the main combustor.

(U) Both precombustor and gas-generators may require a complicated injection system to maintain flow stability over the wide operating box. The XLR129P-1 chambers have thus far demonstrated a 5:1 throttling capability with a fixed geometry injection. There is no such body of data on a throttling gas-generator.

TURBINE DRIVE GASES

A/S TAP-OFF

- CUT & TRY
- FUEL BIAS
- MULTI-MANIFOLD
- UNKNOWN GAS PROPERTIES

J2-S BACKGROUND

HiPc BELL PRECOMBUSTER:

TURBINE INLET TEMP. & PROFILE (2350^{0}R)

NOT EFF. CRITICAL IN THROTTLING MODE
(CYCLE ADVANTAGE)

250 K UNIT DEMONSTRATED
5:1 THROTTLING

AEROSPIKE TAP-OFF AND HI PC BELL PREBURNER TEMPERATURE DATA (U)
(Summary: Basic Technology Issues)

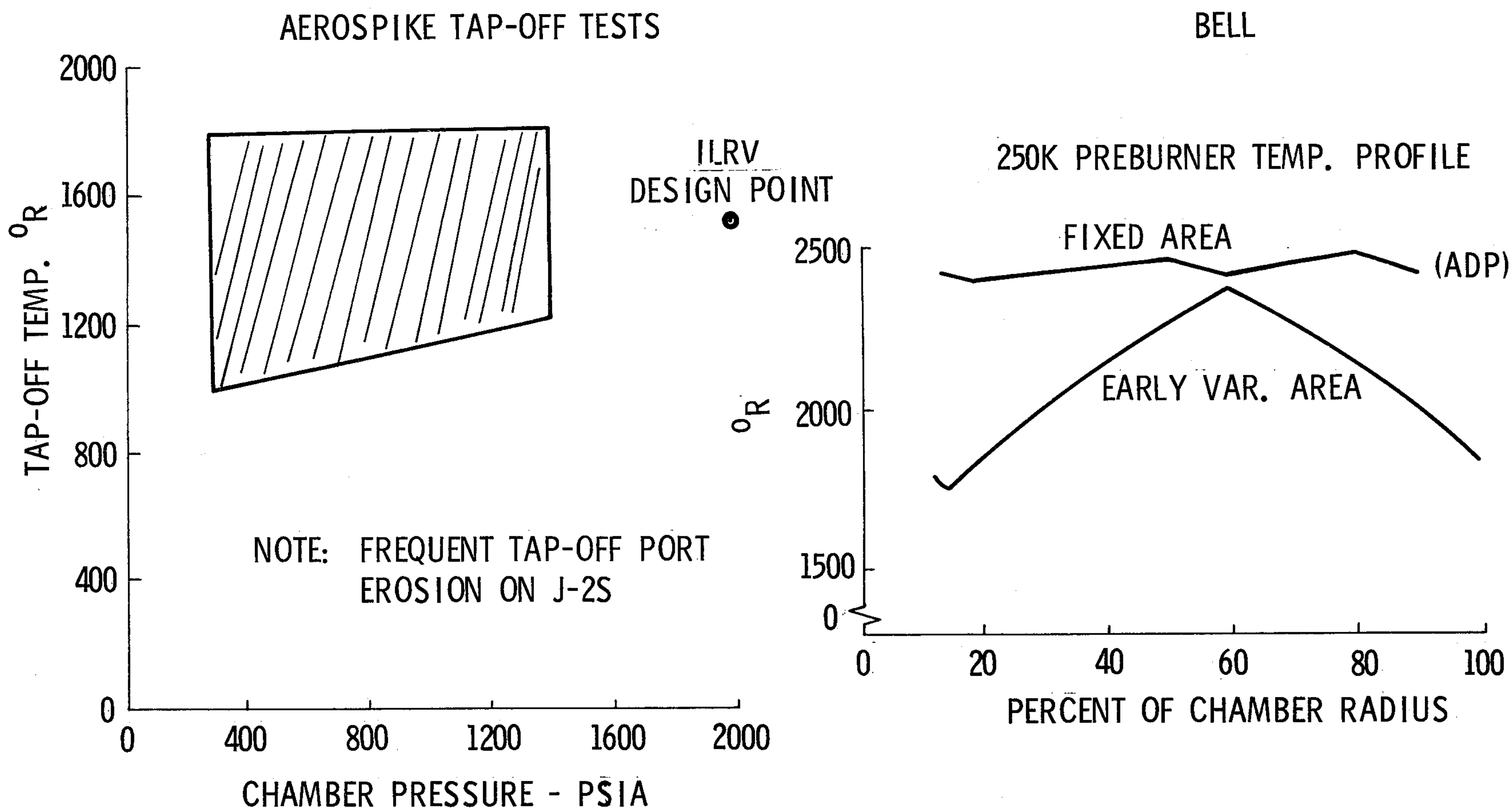
(U) The left side of this chart shows available LO_2/LH_2 tap-off gas temperature data compared with that required for the Space Shuttle Engine. Most of these data were obtained during the J-2S* exploratory programs.

(U) The status of LO_2/LH_2 preburner development is also displayed. For the RREP, early designs of variable area injectors were discarded for that of a simpler fixed geometry system using dual orificed LO_2 injectors. These have demonstrated the target performance of 2350°R with less than ± 150° profile variation over a 5:1 throttling range. This test background does not infer that 10:1 throttling will be cheaply obtained, but that it can be confidently obtained in the precombustor with or without variable injector geometry.

*A simplified version of the J-2

THIS CHART WAS TAKEN FROM A JUNE 1969 MSFC PRESENTATION TO DR. VON BRAUN

AEROSPIKE TAP - OFF TEST/ PREBURNER TEMP. PROFILE DATA



SOURCE MSFC

STRUCTURAL, THROTTLING, AND IDLE MODE ISSUES (U)

(Summary: Basic Technology Differences)

(U) The structural issues center about the packaging and ducting of the staged combustion (S/C) cycle as typified by the Pratt and Whitney transition case concept and the type of structure necessary for an aerospike (A/S). A transition case is a highly loaded structure which must support the major engine components and contain the high pressure turbine exhaust gases. In the XLR129P-1 program, the primary concern is that of H_2 embrittlement of nickel alloys. The A/S requires lightweight high pressure structures for both inner and outer sides of the chamber; plus a larger amount of high pressure ducting and manifolds. The inner chamber surface must withstand severe buckling type loads. This is to be compared with a conventional high pressure cylindrical chamber and simpler plumbing of a bell engine. This comparison further diverges with consideration of a dual combustor having three chamber surfaces, the center of which must be able to withstand load reversals. With the knowledge available, the A/S fabrication problem must be judged more severe than a S/C bell engine.

(U) The throttling comparison also favors a S/C cycle since its control requirements are similar to turbojets. For thrust control the precombustor LO_2 flow is varied. Mixture ratio (MR) is obtained by controlling turbine gas distribution. Fixed geometry injectors should be satisfactory for combustion, but at most the precombustor will require variable area. The tap-off A/S will probably require two tap-off hot gas control valves (for thrust and MR) plus main chamber liquid control valves. The gas-generator A/S will have controls for both combustors and require a turbine bypass. (The original objective of the tap-off J-2S was to simplify the operation of the gas-generator J-2). It is possible but not probable that fixed geometry injectors will suffice for the A/S since high performance (low pressure drop) is required over the operating range.

(U) Idle-mode (pressure-fed operation) is relatively simple with a S/C cycle. LH_2 flows into the precombustor, through the turbines where no power is extracted and into the main chamber where it is ignited and burned with LO_2. Pratt & Whitney calculations indicate that 1% thrust at MR of 2.0 with LH_2 at 45 psia may be possible. A satisfactory scheme for tap-off A/S idle-mode has not been proposed. For the gas-generator A/S it is proposed that the gas-generator be operated at a MR of 1.0 and thrust be obtained as base bleed. Rocketdyne states this could provide 0.2% thrust. The S/C idle-mode performance should be at least 20 seconds higher Isp than the gas-generator A/S due to MR effects alone and has the greater thrust output potential.

STRUCTURAL, THROTTLING, & IDLE MODE ISSUES

	AEROSPIKE	HiPc BELL
STRUCTURAL:	OART CHAMBER PROGRAM DUAL COMBUSTION CHAMBER	COOLED TRANSITION CASE (NOT TIED TO COMB. DEVICE)
THROTTLING:	???	CYCLE ADVANTAGE XLR129P-1 5:1 COMPONENT
IDLE MODE:	STUDIES SHOW DESIRABLE	
	TAP - OFF HOW ??	MAX M. R. OPERATION >2.0
	GAS - GENERATOR	SIMPLE FLOW SCHEME
	0.2% THRUST M. R. ≈ 1.0	1% THRUST
	DUCT INTO BASE	>20 SECONDS HIGHER I_{SP}

SPACECRAFT CENTER OF GRAVITY (U)

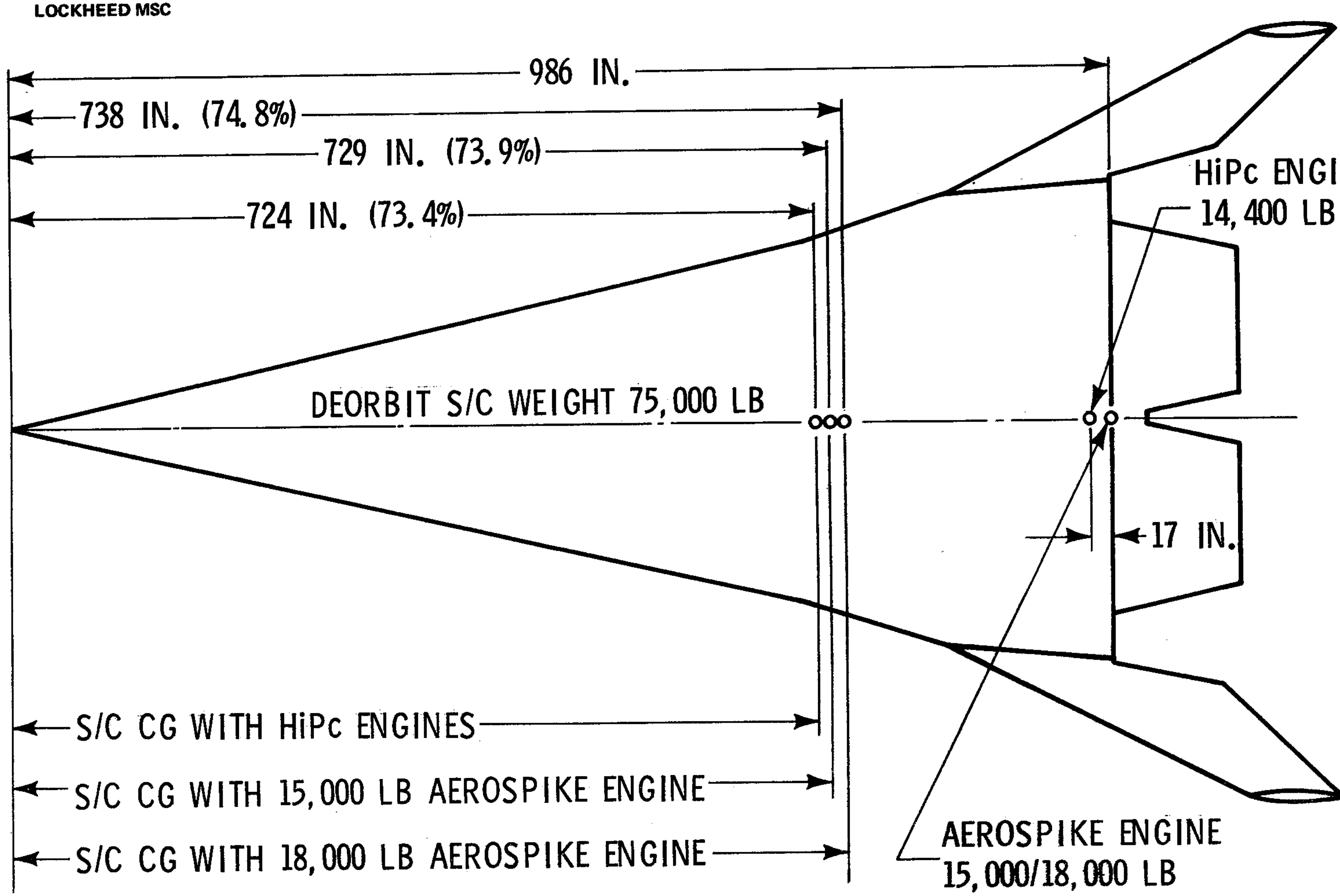
(Summary: Basic Configuration Issues)

(U) The further aft the spacecraft center of gravity, the more difficult it is to trim. The Lockheed stage-and-a-half orbiter configuration shown represents a "worst" case of engine effect on spacecraft c.g. because all the propulsion weight is carried in one stage from lift-off to landing. Lockheed, using engine contractor data, found a small but insignificant advantage for the staged combustion (Hi Pc) engines over the aerospike engines.

(U) It is therefore assumed that engine induced c.g. effects are not germane to engine system selection.

SPACECRAFT SENSITIVITY TO ENGINE CG (DEORBIT CONDITION)

SOURCE:
LOCKHEED MSC



INTEGRATED AEROSPIKE PROPULSION FOR SPACE SHUTTLES (U)
(Summary: Basic Configuration Issues)

(U) The next three charts show early concepts for vehicle integrated aerospike engines. Shown are Convair (14b), Lockheed (14c), and McDonnell Douglas (14d) configurations. The configurations are a Convair 2-Module semicircle, the Lockheed 2-dimensional, 5-Module linear strip, and the McDonnell Douglas 60-degree 6-module engine with dual combustor and illustrate the differences in aerospike shapes proposed to optimize vehicle performance.

50

(U) It is evident that aerospike integrated engine designs and base configurations may be quite different in the event two or more vehicle contractors are funded for the shuttle; each with his own vehicle concept. As such, base design will have to be more nearly complete before the integrated segment/module hardware development can be initiated.

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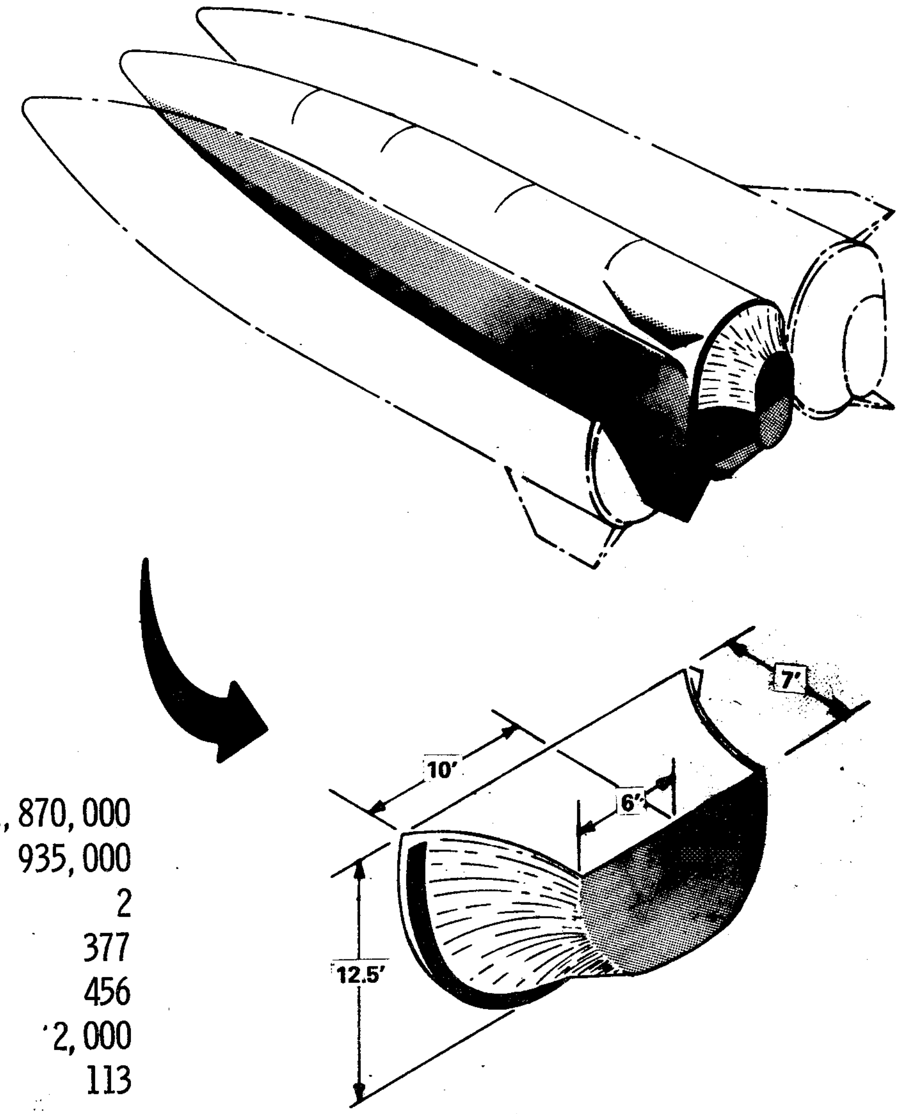
INTEGRATED PROPULSION FOR GENERAL DYNAMICS CONVAIR DIVISION'S ILRV CONFIGURATION (U)

SOURCE:
ROCKETDYNE
A DIV. OF NAR



TOTAL THRUST (VAC.), POUNDS	1,870,000
ENGINE MODULE THRUST, POUNDS	935,000
NUMBER OF MODULES	2
SPECIFIC IMPULSE (S.L.), SECONDS	377
SPECIFIC IMPULSE (VAC.), SECONDS	456
CHAMBER PRESSURE, PSIA	2,000
EXPANSION AREA RATIO	113

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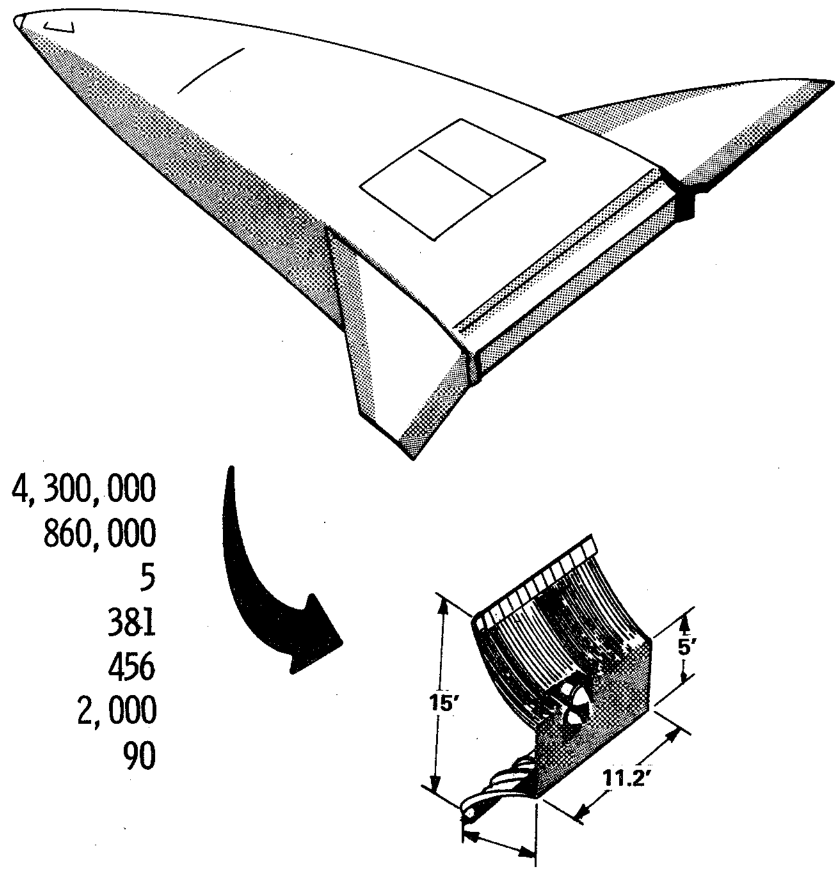
INTEGRATED PROPULSION FOR LOCKHEED SPACE SYSTEM DIVISION'S ILRV CONFIGURATION

SOURCE:
ROCKETDYNE
A DIV. OF NAR



• TOTAL THRUST (VAC.), POUNDS	4,300,000
• ENGINE MODULE THRUST, POUNDS	860,000
• NUMBER OF MODULES	5
• SPECIFIC IMPULSE (S.L.), SECONDS	381
• SPECIFIC IMPULSE (VAC.), SECONDS	456
• CHAMBER PRESSURE, PSIA	2,000
• EXPANSION AREA RATIO	90

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INTEGRATED PROPULSION FOR McDONNELL DOUGLAS ILRV CONFIGURATION DUAL CHAMBER (U)

SOURCE:
ROCKETDYNE
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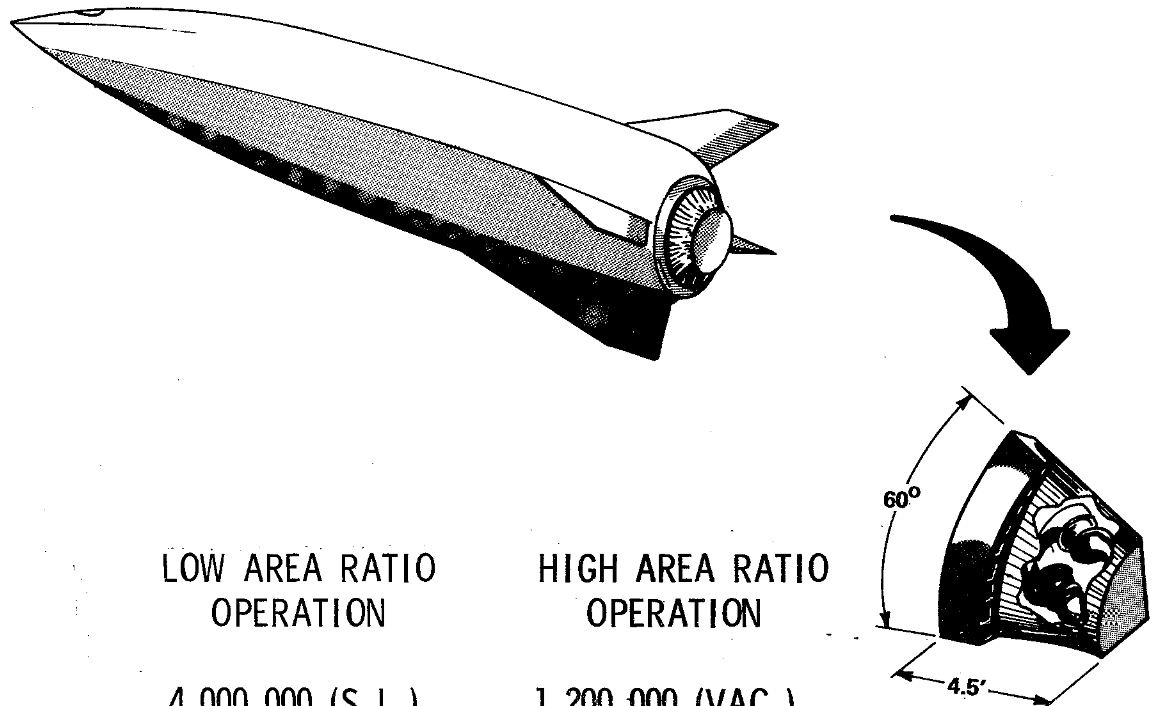

	LOW AREA RATIO OPERATION	HIGH AREA RATIO OPERATION
TOTAL THRUST, POUNDS	4,000,000 (S.L.)	1,200,000 (VAC.)
NUMBER OF MODULES	6	6
SPECIFIC IMPULSE, SECONDS	388 (S.L.)	454 (VAC.)
CHAMBER PRESSURE, PSIA	2,000	2,000
EXPANSION AREA RATIO	23.5	90

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SEP 5 1969

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XLR129P-1 DEMONSTRATOR PROGRAM VALUE (U)
(Summary: Development Issues)

(U) The value of the present Air Force program to a space shuttle engine is shown. Although addressed to a 1,000K lb thrust engine, the validity of the statements is increased significantly as engine thrust approaches that of the 250K XLR129P-1.

(U) The XLR129P-1 program has finished its design phase, and hardware procurement and testing is underway paced by the available funding. Continued progress will provide confidence in component performance and scaleability to shuttle thrust levels. (Both main combustors and precombustor designs are the results of 8 years of test experience starting at the 10K, then the 50K and now the 250K thrust level.) The 250 K precombustor will be useful as a facility item as it will provide proper temperature and pressure gases for turbine and main chamber developments in conjunction with the available 350 K NASA pumps.

(U) The XLR129P-1 controls will be the prototype for a staged combustion shuttle engine and therefore is of value.

(U) Strictly speaking, testing of XLR129P-1 turbomachinery is not necessary for the shuttle engine development. However testing of 250K XLR129P-1 turbomachinery would verify design and performance and when tested with the combustors in a transition case (Powerhead testing) will confirm the Pratt & Whitney design.

(U) Finally, completed demonstrator engine testing will be of value in providing system dynamics, pogo, lifetime, maintenance, and overhaul data necessary for vehicle design and operation.

VALUE OF XLR 129-P-1 DEMONSTRATOR PROGRAM TO 1 MILLION LBS THRUST ENGINE

- PRECOMBUSTOR
  - 1/4 SCALE INJECTOR PATTERN
  - 50% MAX TEMPGAS $\approx 2350^{0}$R FOR TURBINE MAPPING & LIFE (AT 500,000 LB F - FULL SCALE)
  - 25% GAS FOR MAIN COMBUST. & TP/A
- MAIN COMBUSTOR
  - 1/4 SCALE INJECTION PATTERN (RIGIMESH FLOW CONTROL)
- CONTROLS
  - SAME (SIMILAR) AS FULL SCALE
- TRANSITION CASE
  - VERIFY CONCEPT & COOLING LOOP
- TURBOMACHINERY
  - DEMONSTRATION LIFE, DN, BEARINGS, EFFIC. (350 K NASA PUMPS AVAIL FOR FACILITY USE)
- FULL SCALE 250 K DEMONSTRATION
  - VERIFY SYSTEM DYNAMICS: DIAGNOSTICS & MAINTENANCE SCALE VEHICLE

XLR129P-1 DEMONSTRATOR HARDWARE AVAILABLE (U)

(Summary: Development Issues)

(U) The figure shows the XLR129P-1 hardware available for testing as of September 1969. The XLR129P-1 is essentially entering its test phase and hopefully will shortly demonstrate component performance and life. Design work on all "Demonstrator" hardware is essentially complete and, paced by the available funding, is scheduled for manufacture and test. For example fuel turbopump testing is scheduled for the 4th quarter of CY69 and oxidizer turbopump testing for 1st quarter CY70.

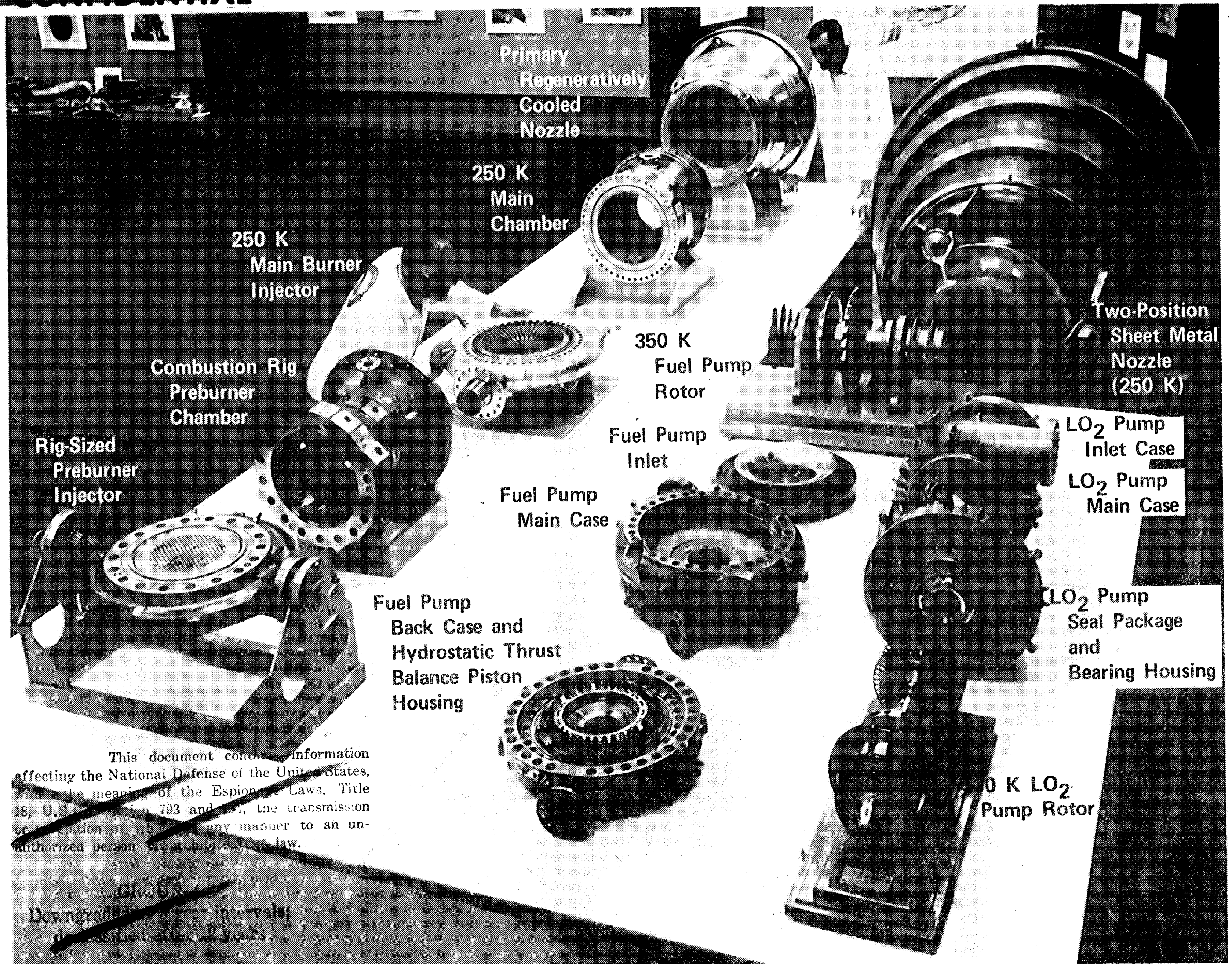

Figure 15b

FC 19169
691308
GS 12440

XLR129P-1 DESIGN FOR MAINTENANCE (U)
(Summary: Development Issues)

(U) This chart was conceived by Pratt & Whitney to show the "design for maintenance" principle in which the major engine components are to be bolted and sealed to the transition case with gaskets. The intent is to provide inspection and maintenance capability into a simple disassembly procedure.

(U) The chart also illustrates the independent component philosophy of the cycle. Derated components can be assembled into an engine allowing for early engine system testing and continued performance improvements and lifetime (TBO) demonstrations. The cycle characteristics and technique of assembly assure that component improvements are carried early into the engine system. However engine system test problems will probably not require component performance requalifications.

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XLR129 DESIGN FOR MAINTENANCE (U)

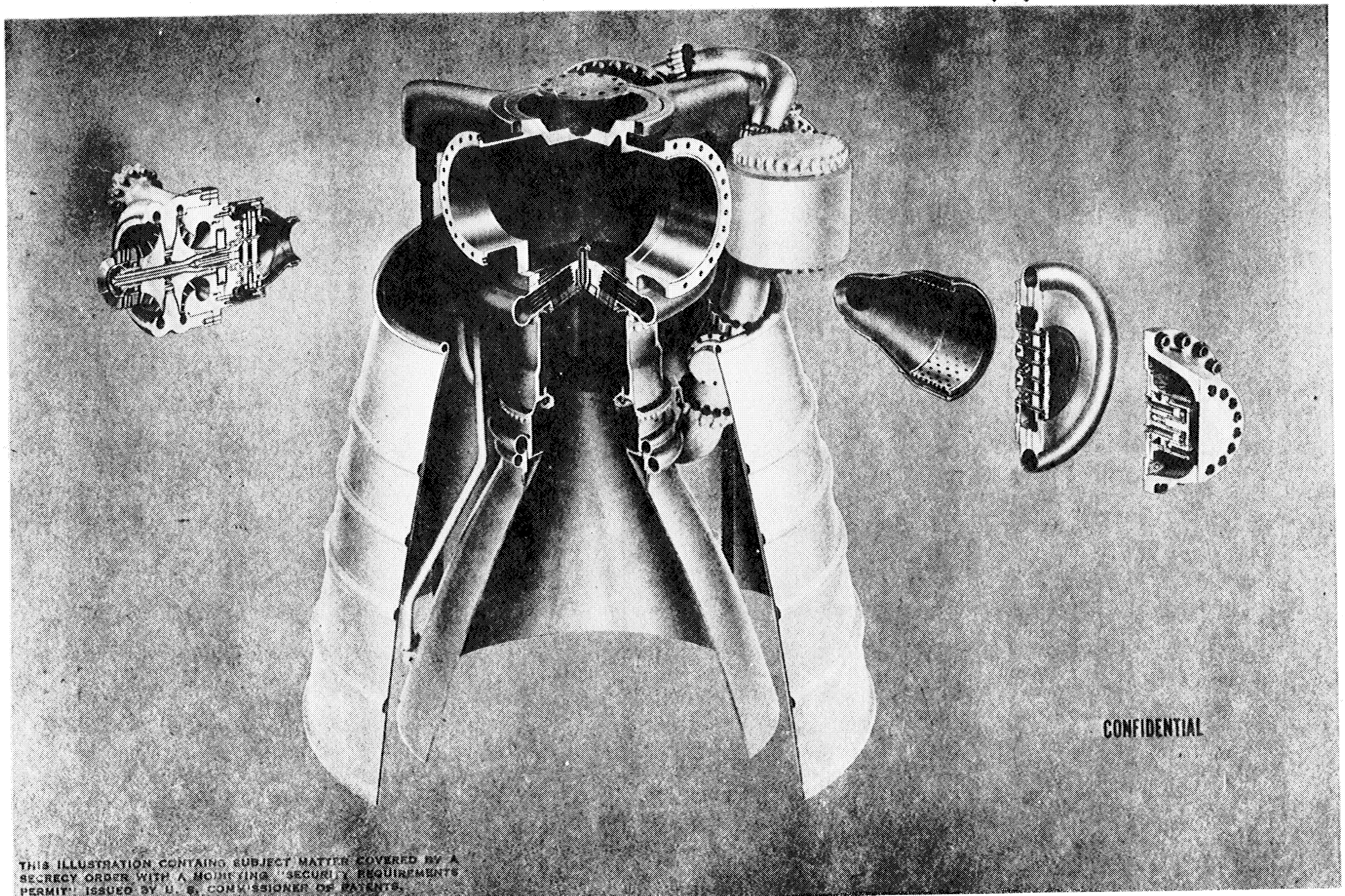

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Figure 15c

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